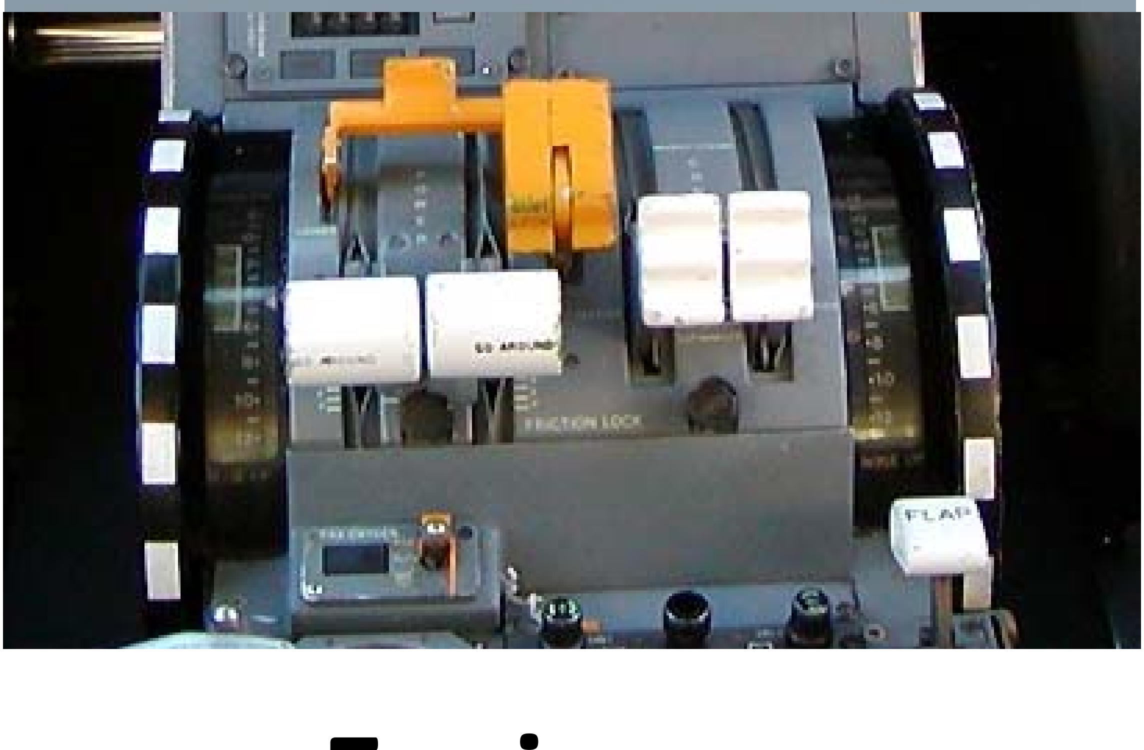
EMBRAER 120



Engines

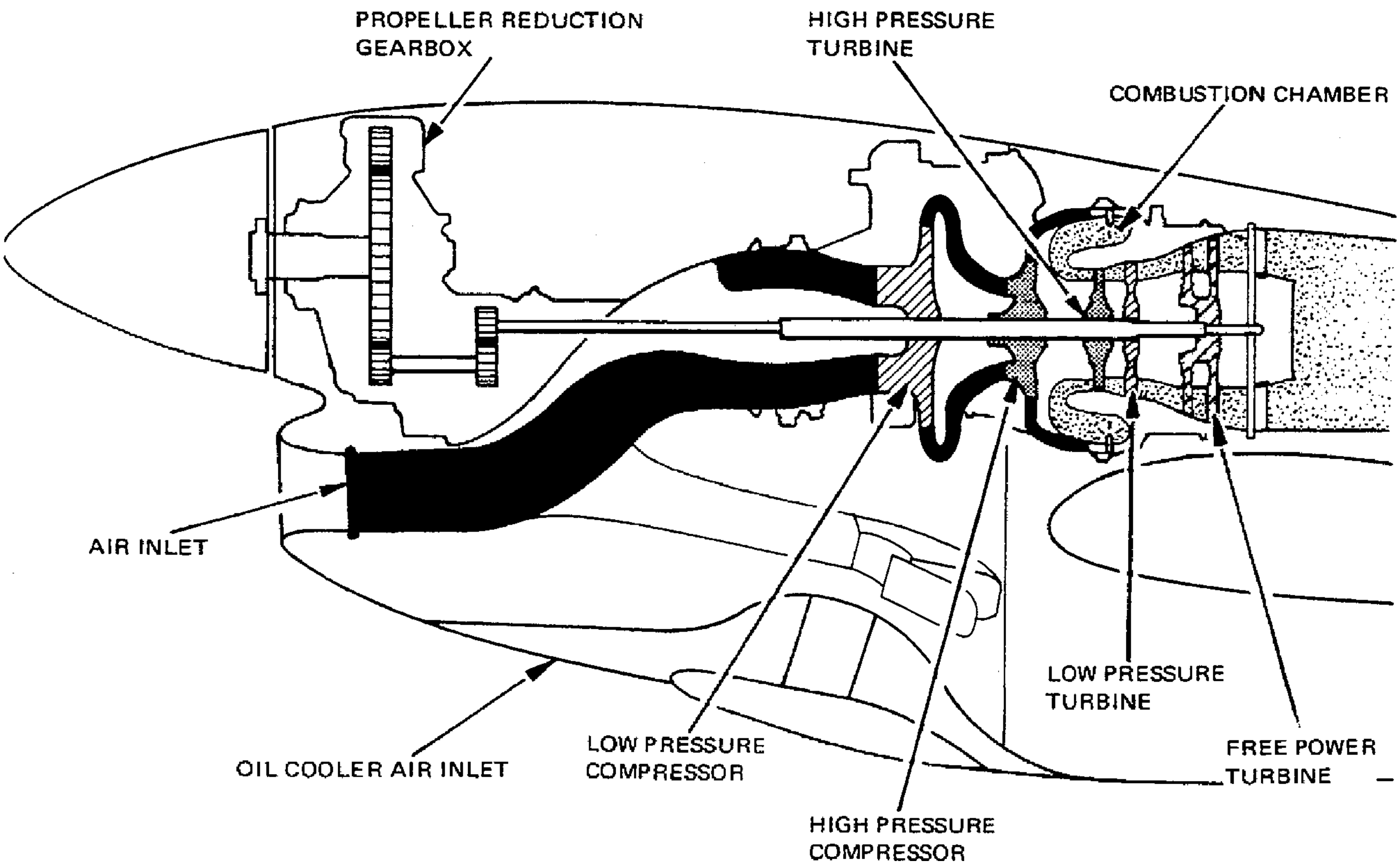
GENERAL DESCRIPTION

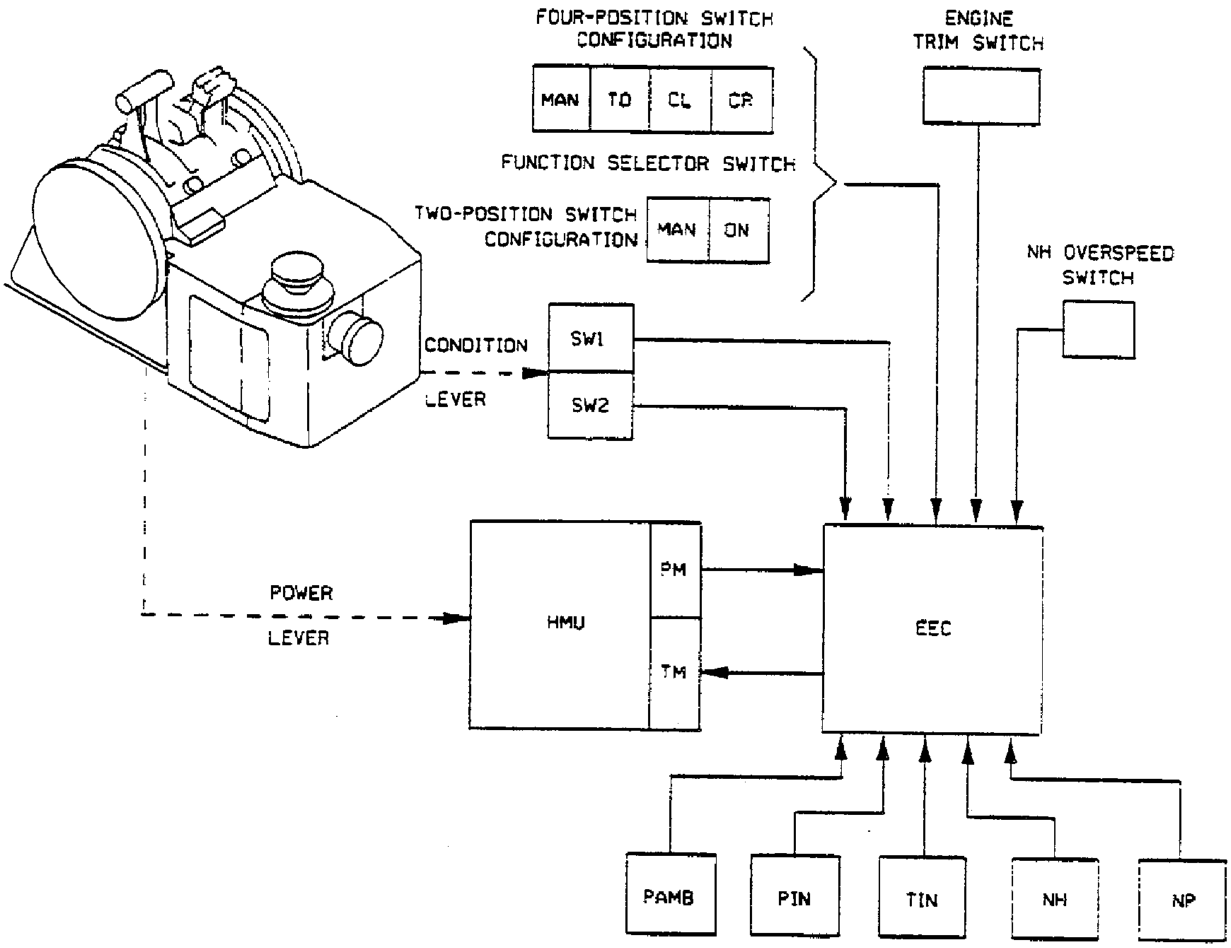
The EMB-120 is powered by two Pratt & Whitney turbopropeller engines which may be either the PW 118, PW 118A or the PW 118B models.
Maximum take-off power rating of the engine models is given in the table below:

Engine Model	PW 118	PW 118A	PW 118B
SHP (KW)	1800	1800	1800
Flat rated at °C (°F)	33.0 (91.4)	42.1 (107.8)	44.9 (112.8)

ENGINE FEATURES

The engine comprises two independent modules, the TURBOMACHINERY (TMM) and the reduction gearbox. The turbomachinery comprises a twin-spool compressor, Low Pressure Compressor (LP) and High Pressure Compressor (HP), a low pressure turbine, a high pressure turbine and a two-stage power turbine. The turbomachinery produces power through compression, combustion, and turbine rotation. The reduction gearbox reduces the input speed from the torque shaft of the turbomachinery (20000 RPM) to the propeller shaft (1300 RPM).
The low-pressure and high-pressure centrifugal compressor provide two stages of compression. Compressed air enters the reverse-flow combustion chamber, the fuel manifold and nozzles supply metered fuel, and two spark ignites located in the combustion chamber ignite the fuel-and-air mixture. The hot expanding combustion gases are directed towards the turbines.
The engine is a "Free turbine" design featuring three independent shafts, each rotating at an individual speed. Each vane and corresponding turbine converts energy from the combustion gas into a mechanical rotational force. The high-pressure turbine turns the high-pressure impeller and accessory gearbox. The low-pressure turbine turns the low-pressure compressor. The two-stage power turbines provide rotational force for the reduction gearbox.
During the start sequence, the high-pressure compressor is rotated by an electric starter motor mounted on the accessory gearbox. When the engine is running, the HP compressor rotates accessories such as the fuel and oil pumps mounted on the accessory gearbox.
A hydromechanical fuel control unit (HMU) regulates fuel flow to the fuel nozzles in response to power requirements and flight conditions. The HMU can be operated in EEC mode (electronic engine control) or Manual Mode (manual pilot input).
The engine also provides bleed air for the pressurization and de-ice systems.


PROPELLER REDUCTION GEARBOX
HIGH PRESSURE TURBINE
COMBUSTION CHAMBER
AIR INLET
LOW PRESSURE TURBINE
FREE POWER TURBINE
OIL COOLER AIR INLET
LOW PRESSURE COMPRESSOR
HIGH PRESSURE COMPRESSOR



- - - - - MECHANICAL LINKAGE

———— ELECTRIC CONNECTION

CL - CLIMB
CR - CRUISE
EEC - ELECTRONIC ENGINE CONTROL
HMU - HYDROMECHANICAL METERING UNIT
MAN - MANUAL
NH - HIGH-PRESSURE ROTOR SPEED
NP - PROPELLER SPEED
PAMB - AMBIENT PRESSURE
PIN - ENGINE INLET AIR PRESSURE
PM - POTENTIOMETER
SW1 - SET HIGH/LOW NP FUEL GOVERN SW
SW2 - NP FUEL GOVERN CANCEL SW
TIN - ENGINE INLET AIR TEMPERATURE
TM - TORQUE MOTOR
TO - TAKEOFF

EEC INPUT/OUTPUT SIGNALS SCHEMATIC

In the event of EEC reversion to the manual mode the MANUAL white light in the EEC control panel and the EEC red light in the glareshield panel will illuminate.
When the EEC is turned ON, the initialization procedure will be accomplished, and will take a short period of time. If the airplane altitude is above 14000 ft, during this short period of time the engine fuel flow will be initially reduced and then increased in order to reach the programmed value within the EEC schedule.
This fact may cause an undesirable engine parameters fluctuation.
After the initialization period, the EEC will normally assume the engine fuel control.
In order to prevent these fluctuations proceed as follows:

- Perform the EEC transition from manual mode to normal mode during climb at the lowest convenient altitude.
- Reduce power lever to a lower power setting before switching the EEC on.
- After the stabilization of engine parameters, advance the power lever to obtain the desired power setting.

NOTE: This procedure shall be performed separately on each engine, if both EECs require reset/transition.

ELECTRONIC ENGINE CONTROL (EEC)

The EEC is a microprocessor-based engine control which executes a program defined by the programmable read-only memory (PROM) and it is mounted on the engine air inlet front case, at nine o'clock position.

The EEC has the function of controlling the engine fuel flow within the EEC schedule upper and lower limits established in the HMU.

EEC may be either on or off to start the engine. EEC on startings will be characterized by two temperature peaks, while the EEC off is characterized by slower startings and only one temperature peak.

It is important to observe that below 25% N_H, only the HMU is responsible for controlling the fuel flow. Above 25% N_H, the EEC, when on, begins to govern the minimum N_H or N_P speed fuel flow to the engine.

For this control function, the EEC has stored in its memory, curves of high pressure rotor speed (N_H) as a function of PLA (power lever angle) and propeller speed (N_P) as a function of PLA.

The EEC acquires and converts input signals to digital data, evaluates the validity of the data, performs the necessary computations and then outputs the proper control signals to maintain the engine within specified parameters.

After processing the input signals, the EEC provides the following output signals:

– Current signal for the actuation of the torque motor. With this signal, the EEC changes the fuel flow.

– Signal for the HMU reversion from normal mode to manual mode with the consequent inhibition of the torque motor.

In the event of EEC electrical power interruption, or a particular sensor failure and software malfunction, the control system will revert to the HMU schedule, defined as manual control mode.

This reversion is characterized by the following:

1. Loss of engine power (in case of electrical power interruption or major failure), recoverable by advancing the power lever.
   In case of a particular sensor failure without loss of electrical power, the EEC will keep the engine power constant.
   This is achieved by the EEC fail-fixed function, which allows the EEC to maintain the engine power constant during the reversion to the manual mode.
   The EEC freezes and keeps constant the last current signal sent to the torque motor. This function is effective below 14000 ft.
   On aborted takeoffs this function will enhance reverse thrust asymmetry. The fail-fixed function is canceled when the EEC is reset or turned off.
2. Power lever stagger when setting equivalent torque on the engine in manual mode relative to the other in normal mode.
3. Slower than normal acceleration, at lower altitudes.
4. Loss of fixed throttle climb feature.

CONDITION LEVERS

The Condition Lever controls the propeller speed and the fuel opening for the engine start.
The following discrete condition lever positions are provided at the control stand:

- MAX RPM (Maximum RPM);
- MIN RPM (Minimum RPM);
- FEATHER;
- FUEL CUT OFF.

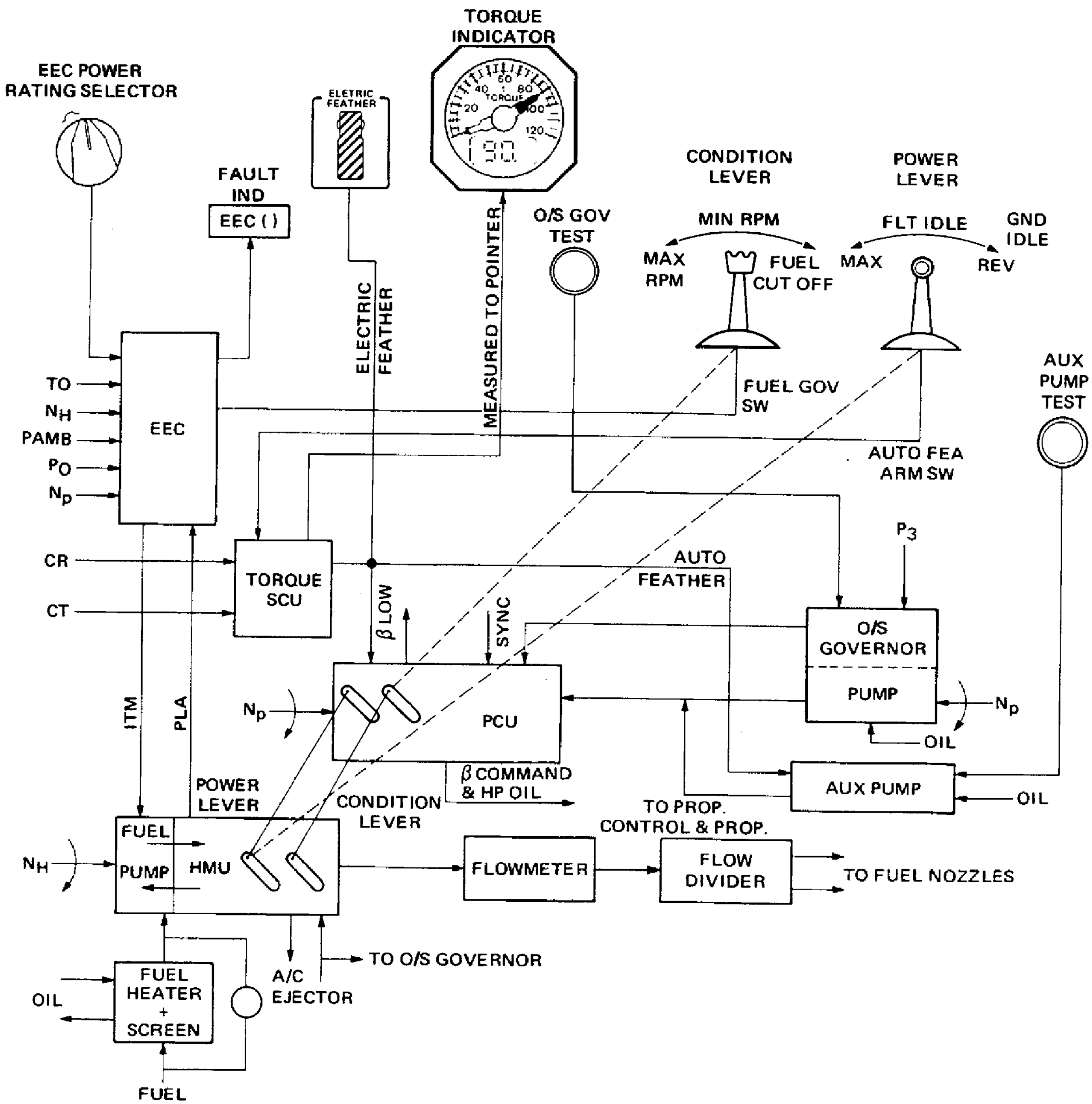
A gate is provided at the MIN RPM position. To move the condition lever further backward, its backstop release levers must be pulled upward. A backstop is provided in FEATHER position: To move the condition lever further backward, the backstop release lever installed at the back of the condition lever must be pressed forward.

CAUTION: ABOVE 12000 FT, A FAST MOVEMENT OF THE CONDITION LEVER TOWARD MAX RPM MAY LEAD TO A COMPRESSOR STALL, IF THE POWER LEVER IS NOT AT FLT IDLE.

GUST LOCK

Prevents power lever movement forward of the FLT IDLE position, when set in the LOCKED position. It ensures the locking of the aileron and elevator controls.

CONTROL SYSTEM SCHEMATIC



POWER PLANT SYSTEM CONTROL

The engines have two set of controls, the power levers and the condition levers.

POWER LEVERS

The Power Lever controls the engine power from Reverse to Maximum Power.
The following discrete Power Lever positions are provided at the control stand:

- Max (Maximum Power setting);
- FLT IDLE (Flight Idle setting);
- GND IDLE (Ground Idle setting) and
- REV (Reverse setting).

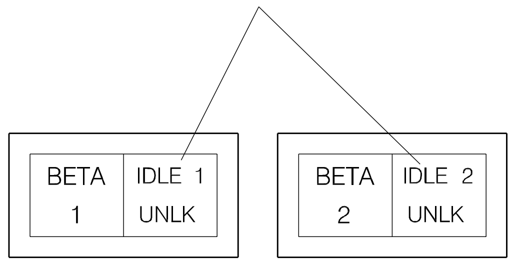
The Power Lever is equipped with two redundant FLT IDLE stops:

- A Primary Stop, provided in the FLT IDLE position;
- A Secondary Flight Idle Stop System (SFIS).

Upon landing, for reverse application, the backstop trigger installed at each Power Lever needs to be pulled upward in order to overcome the Primary Stop (Flight Idle detent) and to move the Power Lever further backward.
The SFIS consists of an electrically actuated solenoid and the related mechanism in each engine nacelle. Whenever the airplane is inflight, the solenoid is energized, locking out the mechanism. Upon landing, the air-ground system switches off the SFIS solenoid, allowing movement of the Power Levers into the GND IDLE and Reverse ranges after the release of the Flight Idle Primary Stops.
Additionally, airplanes Post-Mod SB 120-76-0018 or S/N 120.345 and on are equipped with two independent amber indication lights (IDLE 1 UNLK and IDLE 2 UNLK), which illuminate if the corresponding Secondary Flight Idle Stop System is not operational in flight.

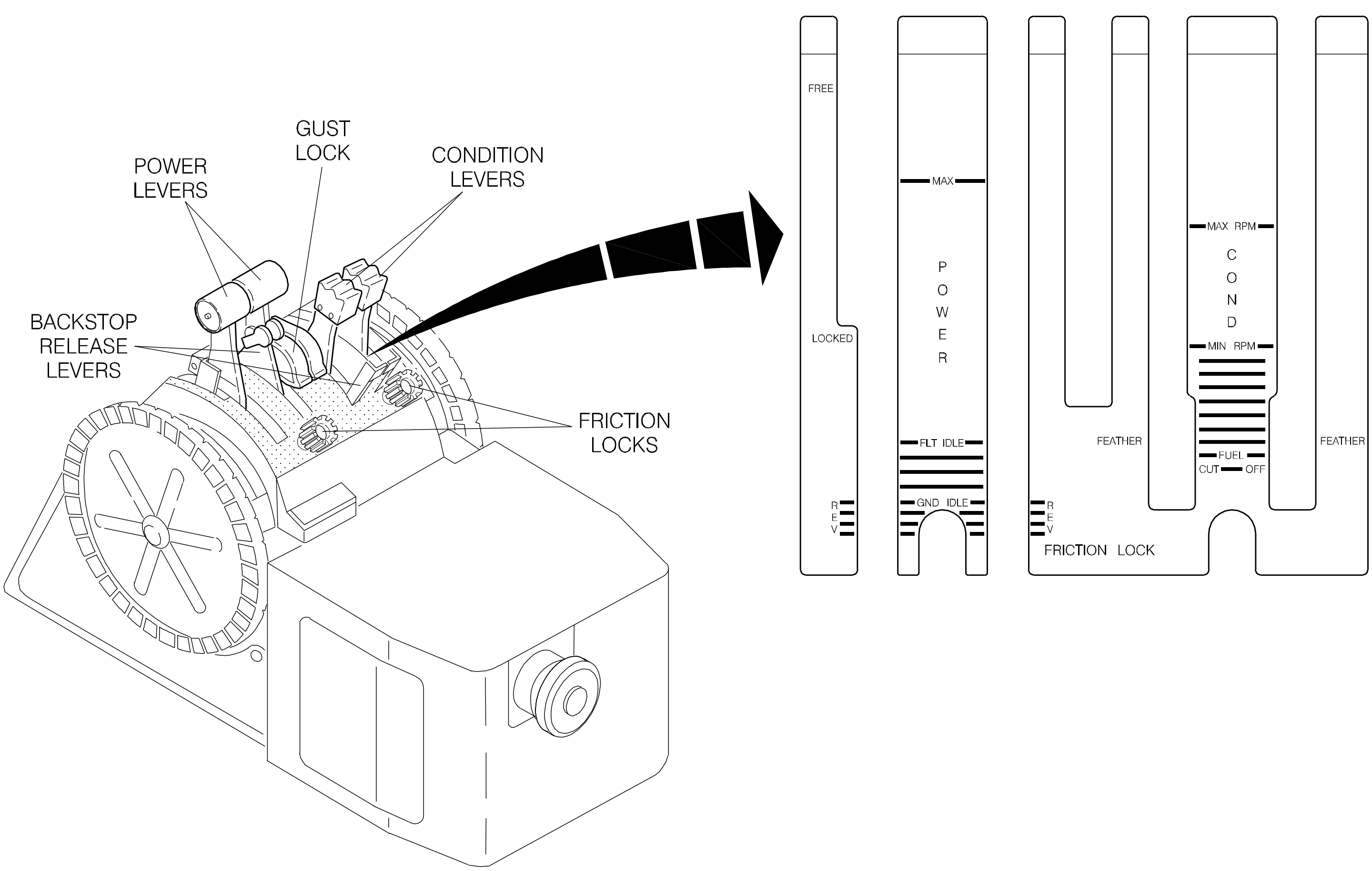
CAUTION:
- NEVER SET POWER LEVER BELOW FLT IDLE IN FLIGHT.
- APPLY REVERSE ONLY AFTER THE NOSE WHEEL IS ON THE GROUND.

IDLE UNLK LIGHTS (AMBER)
ILLUMINATED – When the system is not operational in flight.



(GLARESHIELD PANEL)

Post-Mod. SB 120-76-0018 or S/N 120.345 and on.



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GUST LOCK

Prevents power lever movement forward of the FLT IDLE position, when set in the LOCKED position. It ensures the locking of the aileron and elevator controls.

ENGINE FUEL CONTROL SYSTEM

The engine fuel control system provides metered fuel flow for engine requirements, throughout its operating range.
The engine fuel control is performed by the Hydromechanical Metering Unit (HMU) and the Electronic Engine Control (EEC).

HYDROMECHANICAL METERING UNIT (HMU)

The HMU is installed on engine turbomachinery accessories section and establishes the minimum and maximum limits of fuel flow to be supplied to the engine as a function of the power control lever position (PLA) and high pressure compressor discharge pressure (P_3).
The HMU also incorporates an enrichment solenoid valve which assures an expanded EEC fuel control at altitudes above 14000 ft.
At altitudes below 14000 ft the enrichment solenoid valve is automatically de-energized by the altitude switch.
When the solenoid valve is energized by the altitude switch, it schedules a lower HMU minimum limit fuel flow, allowing the EEC fuel schedule to be followed, being limited only by the maximum and minimum EEC fuel flow limits, without HMU schedule interference.
This fact prevents the occurrence of engine compressor stalls associated with fast power lever transients when operating at high altitudes.
If a failure occurs, and the HMU enrichment solenoid valve remains energized at altitudes below 14000 ft, a 10000 ft baroswitch will trigger the aural warning, the WARNING and EEC red lights.

NOTE: The activation/deactivation of the HMU enrich solenoid may be associated with a slight power transient, which is normal.

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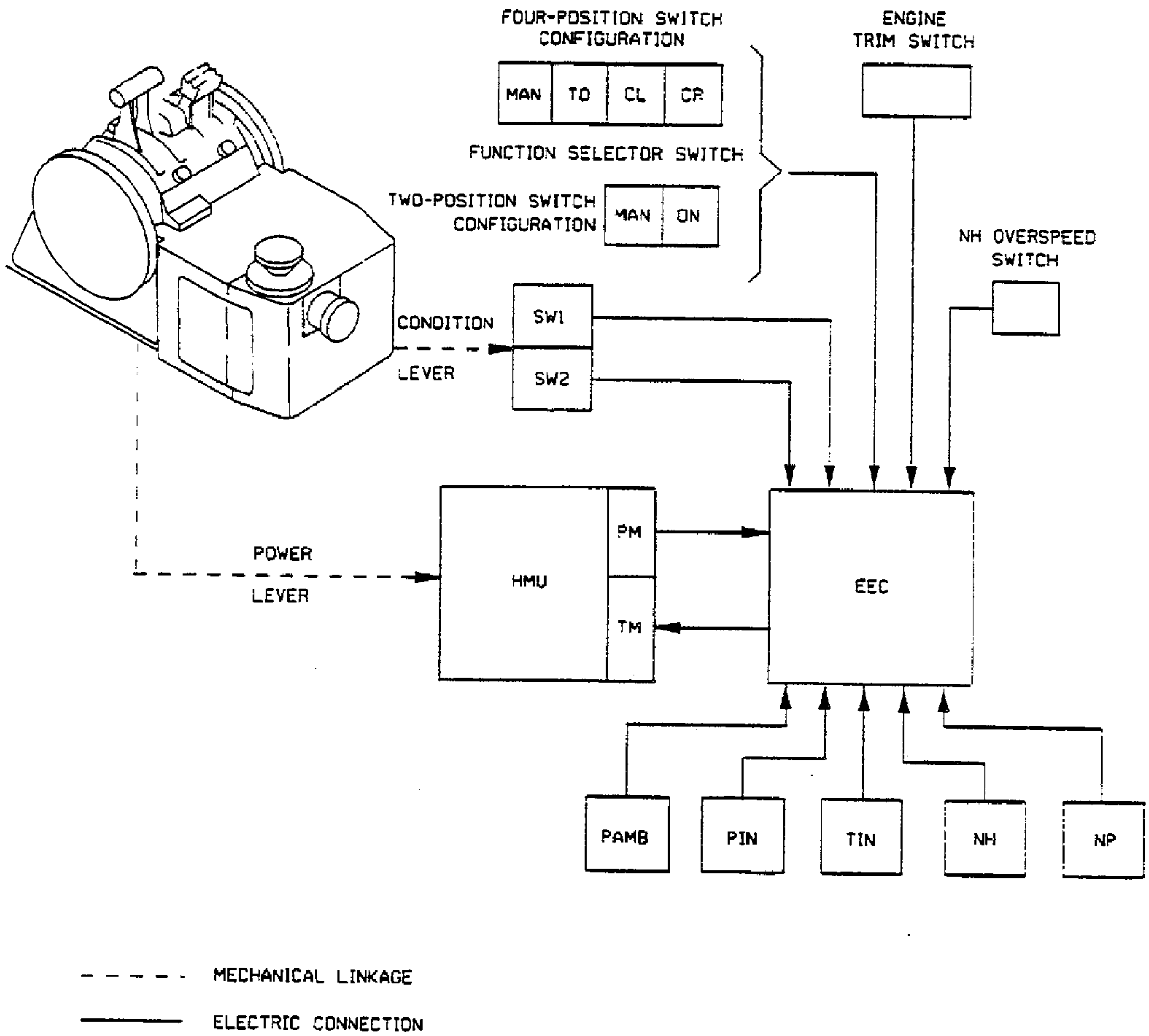

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EEC - ELECTRONIC ENGINE CONTROL
HMU - HYDROMECHANICAL METERING UNIT
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NOTE: This procedure shall be performed separately on each engine, if both EECs require reset/transition.

EEC CONTROLS AND INDICATORS

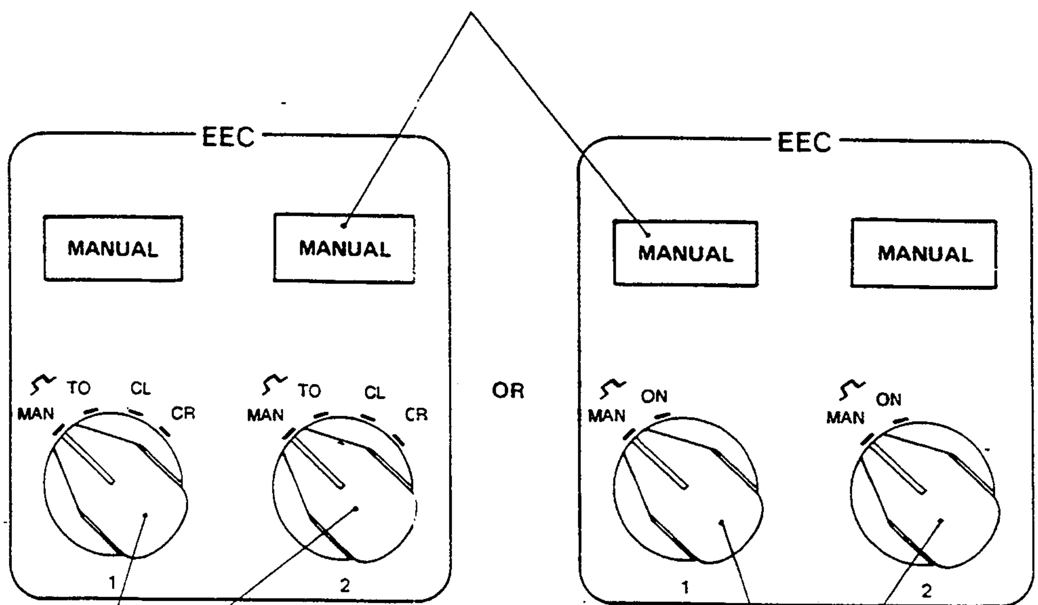
(GLARESHIELD PANEL)

EEC LIGHTS (RED)

ILLUMINATED — When any failure is detected on the relevant EEC or on HMU enrich solenoid valve.

ENGINE No. 1 MANUAL LIGHT (WHITE)

ILLUMINATED — When positioning the EEC power rating selector on MAN position or when EEC reverts automatically to manual mode.



(OVERHEAD PANEL)

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EEC POWER RATING SELECTOR

MAN — Selects the manual mode. EEC is off.
TO — Selects the takeoff condition.
CL — Selects the climb condition.
CR — Selects the cruise condition.

NOTE: Pull knob to change from MAN to TO positions.

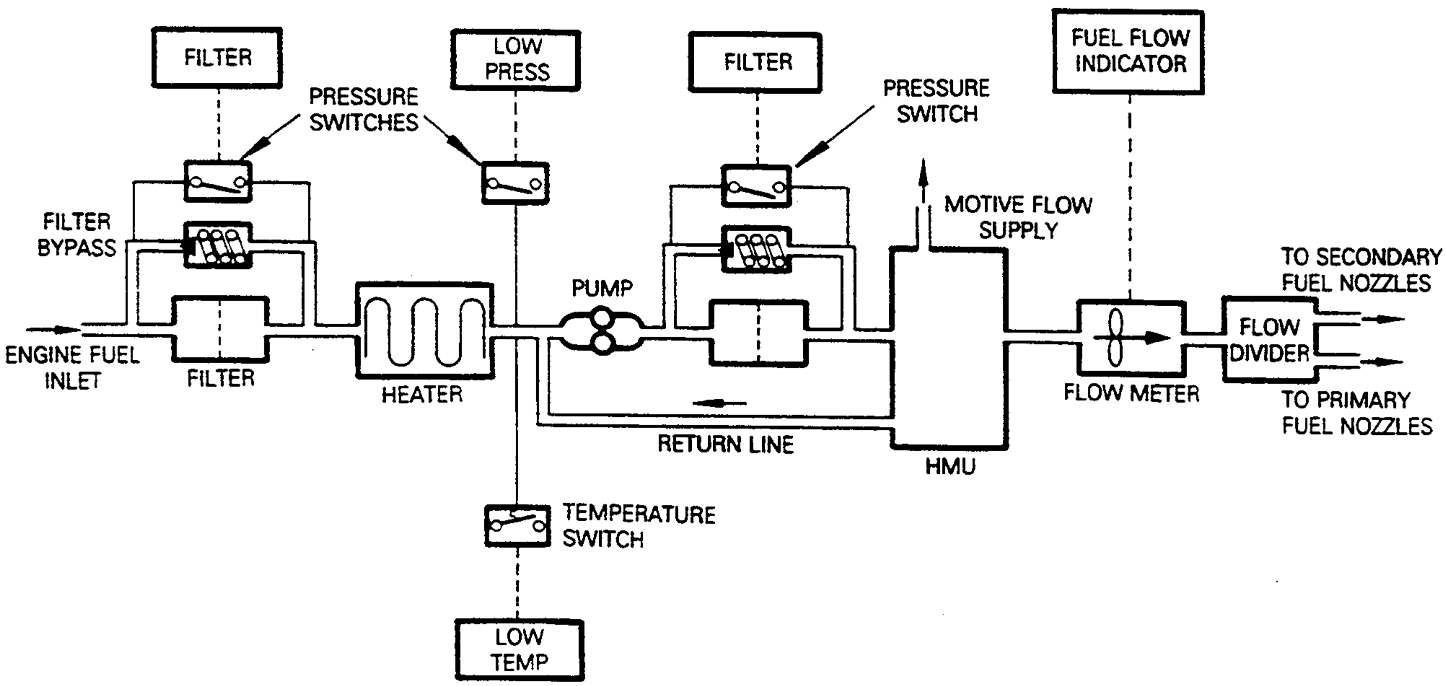
EEC POWER RATING SELECTOR

MAN — Selects the manual mode. EEC is off.
ON — Turn the EEC on.

NOTE: Pull knob to change from MAN to ON positions.

ENGINE FUEL SYSTEM

Fuel is pumped by electric booster pumps or ejector pumps from the aircraft fuel tanks into the inlet of the fuel heater unit. Fuel flows through the inlet filter and the heater element before reaching the fuel pump. The engine-driven fuel pump provides filtered fuel flow for the hydro-mechanical metering unit (HMU). The HMU directs metered fuel to the flow divider, unmetered high-pressure fuel flow to the airframe ejector pumps, and bypass fuel to the fuel pump inlet. The flow divider directs primary and secondary fuel flow to the 14 fuel manifold adapters and nozzles.



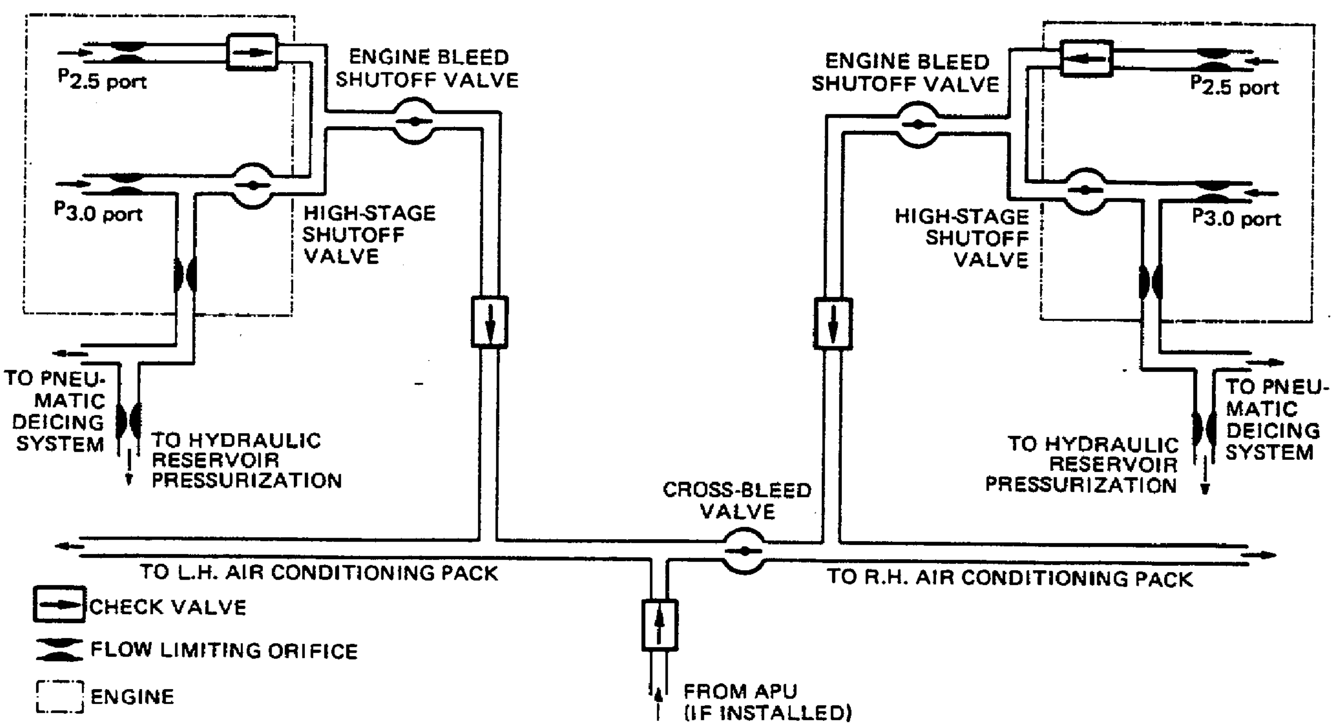
AIR BLEED SYSTEM

Each engine incorporates two compressor bleed ports: one at the low-pressure compressor (the $P_{2.5}$ port) and the other at the high-pressure compressor (the $P_{3.0}$ port). The air serves the purpose of supplying airplane systems, such as air conditioning, pressurization, pneumatic deicing, and hydraulic reservoir pressurization.

The system consists of a combined flow limiting orifice/check valve at the $P_{2.5}$ port and a flow limiting orifice plus shutoff valve at the $P_{3.0}$ port. The bleed ducting from the two ports merge to form a single supply duct to the airplane, interconnected by a normally closed cross-bleed valve.

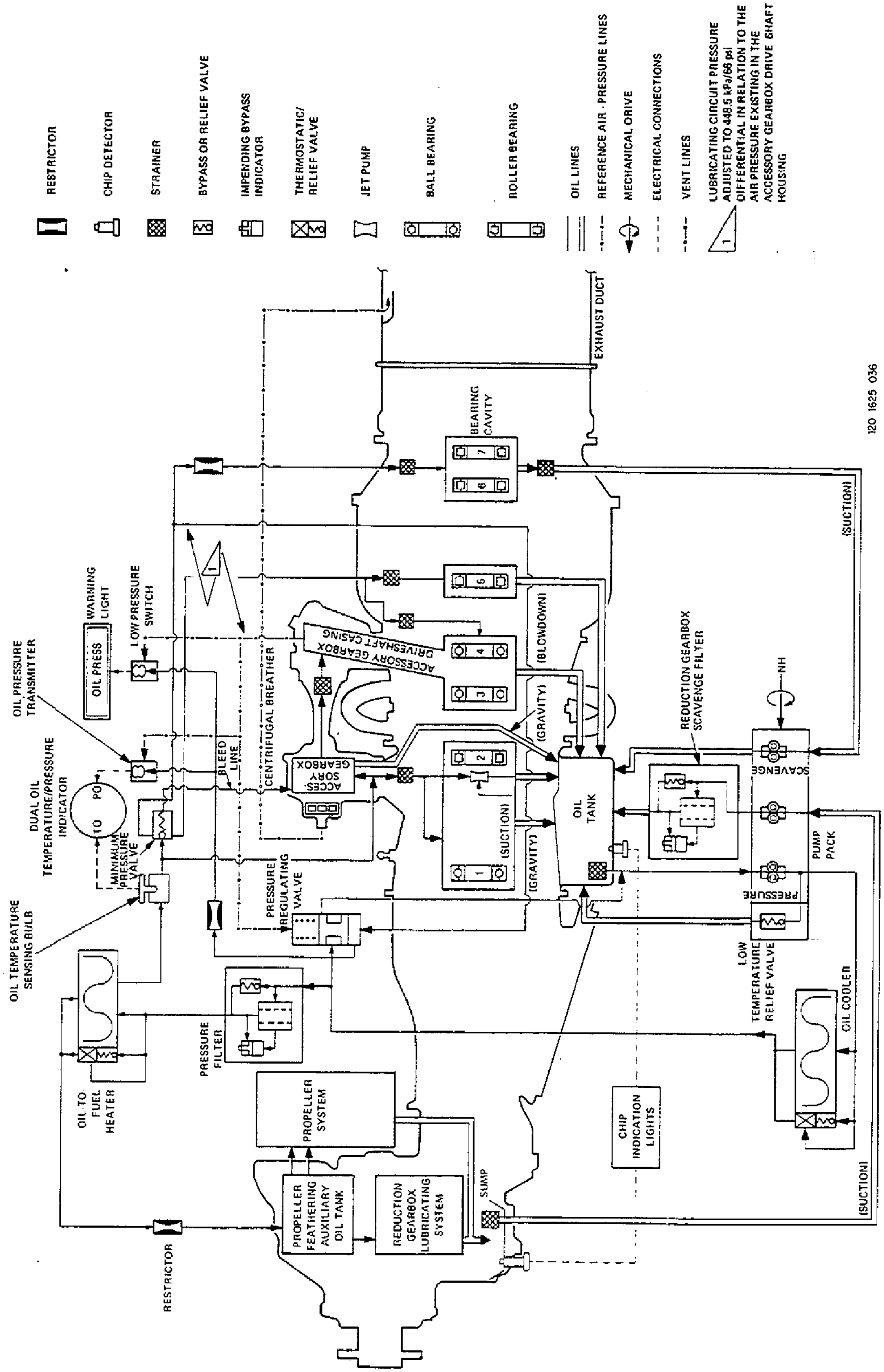
Under operating conditions where the low-pressure compressor delivery pressure is too low (for example, during ground idle or low power descent), the high stage shutoff valve supplies high pressure bleed air to the airplane single supply duct. This condition may sometimes lead to a lower air conditioning efficiency, which may be minimized by setting N_H up to 75%.

Upstream of the high-stage shutoff valve, a tap of the bleed line supplies the pneumatic deicing system bleed line. Air for the hydraulic reservoir pressurization is bled from this line.



ENGINE OIL SYSTEM

The engine lubricating system provides a constant flow of filtered oil under controlled pressure and temperature for lubricating and cooling the engine bearings, reduction gears and gears of the accessory sections of the reduction gearbox (RGB) and turbomachinery modules. The system oil also lubricates and actuates the propeller servo-mechanism.
The lubricating oil is stored in a tank which is integral with the turbomachinery module. The oil level can be easily checked through a sightglass on the left side of the tank.
The oil cooling is performed through an air-cooled oil cooler installed below the inertial separation bypass duct and provided with a bypass for a faster warm-up.
A high pressure gear pump driven by the N_H spool supplies the oil to the system.
A low oil pressure switch sends an electrical signal to the OIL PRESS warning light when the oil differential pressure drops below 40 psid.
A pressure filter with a bypass valve and an impending bypass indication (movable pin) is installed downstream of the pump. A relief valve is provided to prevent damage from pressure surges during cold starts. A fuel heater is installed to cater for engine fuel heating. A minimum pressure valve is fitted upstream of the pressure filter to prevent oil leakage from the bearings, during starts.
Gear pumps scavenge the No. 6 and 7 bearing cavities and the reduction gearbox. All other cavities are scavenged either by blowdown or gravity.
A reduction gearbox filter is fitted downstream of the RGB scavenge pump to protect the turbomachinery and the air-cooled oil cooler against contamination in the event of a gearbox failure.
A movable pin is provided to indicate an impending bypass in the RGB scavenge filter.
Chip detection indication is accomplished through chip detectors and caution lights. A chip detector is installed in the RGB and another in the oil tank. On airplanes Pre-Mod. SB 120-79-0008, the CHIP DETR caution lights are installed on the multiple alarm panel. On airplanes Post-Mod. SB 120-79-0008 or S/N 120.076, 120.079 and on, the engine chip indication lights are installed in the battery compartment for easier maintenance personnel checks.



ENGINE OIL SYSTEM

START/IGNITION

For engine starting, a 28 V DC/400A starter generator drives the high pressure spool (N_H), supplied by an internal 24 V/36 Ah battery or by an external DC power source.
A three-position switch, one for each engine, located on the overhead panel provides the starting control. The start switch has two momentary positions: ON and ABORT.
Selecting the ON position, the starting cycle is initiated and is automatically interrupted as the cutout speed (50% ± 6 N_H) is sensed by the starter/generator. The ABORT position allows interruption of the automatic starting cycle.
An engine-mounted ignition exciter box, powered by the aircraft 28-V DC electrical system, supplies energy to two spark igniters for engine ignition. Normally, the system is used for starting only, but may be used during any other phase of flight, when necessary.
Control of the ignition system is achieved by means of a three-position switch, next to the start switches.
The ignition switch positions are ON-AUTO-OFF. In the ON position, the igniters are continuously energized. In the AUTO position, the igniters are automatically energized every time the engine starting cycle is initiated and are deenergized with the automatic or manual interruption of the starting cycle. In the OFF position, the igniters are not energized, regardless of the starting system operation.
A white IGNITION light, on the start/ignition panel, illuminates, when the ignition system is energized.

INERTIAL BYPASS

The bypass system is the continuous flow type and provides inertial separation protection against bird and foreign object ingestion.
The air outlet is always open, and is sufficient to maintain normal flow to engine and inertial separation protection.